DEVELOPMENT OF A MINIATURE GRIDDED ION THRUSTER

Master Thesis for Erasmus Mundus master program SpaceMaster

Peter Kindberg

Thesis supervisor: Prof. Jaan Praks, Aalto University
Thesis supervisor: Prof. Peter von Ballmoos, Université Toulouse III – Paul Sabatier
Thesis supervisor: Prof. Soheil Sadeghi, Luleå University of Technology

Thesis advisor: D.Sc. (Tech) Antti Kestilä

Abstract

Electric propulsion has matured to the point where it starts credibly to challenge chemical propulsion systems. The high efficiency and larger specific impulse have allowed to lower the weight of the satellites and therefore costs. Even though the thrust of an electric propulsion systems is still just a fraction of that chemical rockets, the possibility of keeping it on for months eventually gives the same amount of momentum.

While CubeSat projects are getting more popular by universities only a few have developed their own miniature ion thruster. These mini thrusters have a thrust force of sub mN to few mN, but they still enable up to 100 kg almost any kind of altering to satellites orbital position and attitude. Another, issue on electric propulsion is the propellant. Xenon is most used propellant fuel in ion thrusters, but because of its high costs, it is out of the reach for many universities.

This study tried tackling into those two topics and create a design of a miniature ring cusp gridded ion thruster. The study wanted to research possibilities of using 3D printing technology and this way simplify the fabrication process. The goal was to create a design which could be used as a foundation for Aalto University's propulsion development and also enable testing of alternative propellants. The propellants studied were iodine, adamantane, and diamantane. All of them are in a solid-state at room temperature, which would be practical for storage. The design is presented at the end of the study.

Co-funded by the
Erasmus+ Programme
of the European Union

Table of Contents

Nomenclature

Symbol	Units	Description
A	A/cm^2K^2	Richardson's coefficient
α		Thrust correction factory
α_m		Mass utilization correction
η_d	eV/ion	Discharge loss
η_E		Electrical efficiency
η_m		Mass utilization efficiency
η_T		Total efficiency
d	m	Distance
ε_0	$m^{-3}\ kg^{-1}\ s^4\ A^2$	Permittivity of free space
F_t	Degrees	Effective thrust angle
g	m/s^2	Standard gravity
I^+		Number of single charged ions
I^{++}		Number of double or more charged ions
I_b	A	Beam current
I_d	A	Discharge current
I_{sp}	s	Specific impulse
I(r)	A/m^2	Ion current density as a function of radius
J	A	Thermic emission
J_i	A/m^2	Ion current density
K	J/K	Boltzmann's constant
M	kg	Ion mass
m_d	kg	Final delivered mass
m_p	kg	Propellant mass
m_i	kg	Initial mass
$\dot{m}_i$	kg/s	Ion mass flow rate
m_p	kg	Required propellant mass
$\dot{m}_p$	kg/s	Propellant mass flow rate
ϕ	J	Work function
P_0	W	Miscellaneous power
P_b	W	Beam power
P_d	W	Power to produce ions
P_{in}	W	Input power
P_T	W	Total power
$\dot{v}$		Coordinate acceleration
v_{ex}	m/s	Effective exhaust velocity
v_f	m/s	Final velocity
v_i	m/s	Initial velocity
Δv	m/s	Change in velocity
T	K	Temperature

1 Introduction

Electric propulsion has matured to the point where it starts credibly to challenge chemical propulsion systems. The high efficiency and thus specific impulse have allowed to lower the weight of the satellites and therefore costs. Even though the thrust of an electric propulsion systems is still just a fraction of chemical rockets, the possibility of keeping it on for months eventually gives the same amount of momentum. NASA concluded in 2007 a test where it had kept their latest electric propulsion device on more than 48 000 hours (5.5 years) [1]. As CubeSats are getting more popular in universities, next natural step is propulsion systems. It is still a challenge to make a CubeSat sized electric propulsion system, mainly due to the power requirements, but there are already few working examples of CubeSat sized propulsion systems [2].

Electric propulsion systems are mainly used for station keeping, orbit changes or as a primary propulsion [3]. The reason for using electric propulsion systems is their efficiency. Chemical rockets can get efficiency up to 35 percent, while electric propulsion devices have recorded more than 90 percent efficiencies [4]. The trade-off on the electric propulsion systems is low thrust, typically less than 100 mN. Still, the key features like large specific impulse, reliability and high exhaust velocities fulfill most mission requirements.

Ion thruster development in the early 1970s was started from developing miniature ion thrusters, which worked well for demonstrating the technology. The development of micro thrusters was then ceased while the focus moved to larger thrusters [5]. Miniature thrusters have gained popularity again and they have been developed at least in Germany, United States and Japan [5]. Miniature radio frequency thrusters have been most popular by different institutions, but there has been few microwave discharge thrusters and one cathode discharge thruster as well.

Because of the vastness of the study area, thrusters are not very popular Master’s thesis subjects. There are some literature studies for bachelors’ thesis, but actual ion thruster building is rare. Especially, building a gridded ion thruster is so rare that the author was not able to find a single study of the subject in master’s level. The studies conducted around the subject is typically made by Ph.D. students or by research facilities.

The main studies used for inspiration in this work were Wirz et al [6] study from micro-ion thrusters, which presents a detailed description of the development of a small ion thruster and Boeva, J.J., [7] who built a pulsed plasma thruster as a master’s thesis for the Delft university of technology. Instead of building a whole thruster, more common seems to be focusing on a single component. Goebel and Chu’s study of lanthanum hexaboride hollow cathodes gave a good insight on hollow cathodes in overall. Also, Farnell’s, massive study of ion thruster optics [8] helped in designing the grid system.

While the original goal of the study was to design, build and test a gridded ion thruster and same time also test alternative propellants, the goal was required to change to find how ion thrusters are designed, what are the main features and how they should be taken into consideration when creating the design. In the end, these findings were used to create a design. The study is also going to go through how a thruster works and explain the working principle of the main components. Another important aspect this thesis wanted to study was propellants. At the moment, Xenon is most common propellant used in ion thrusters. The price of Xenon is high so some alternative propellants are presented with their benefits and drawbacks. Since the study area is so vast and the time limit is rigid, the design is built by using values received from other similar studies. Also, because the page number of the thesis is also limited to 30 pages the study focuses mainly on theoretical design features of ion thruster, without numerical analysis.

1.1 Introduction to electric propulsion

The idea of electric propulsion dates back already to the start of 20th century, it was visualized by two individuals, American Robert Goddard and Russian Konstantin Tsiolkovsky, on the other sides of the planet, nearly same time. It was 1911 when self-taught Russian school teacher and free time rocket scientist, Konstantin Tsiolkovsky introduced the concept of electric propulsion in his publication [9,10]. The introduction of the concept was dated on the time when many physicists were working with cathode rays, is most likely not a coincidence.

Physicist's of that time already knew that the cathode rays had thousand times larger velocities when compared to particles moving in normal temperatures. Tsiolkovsky, as a man who had several years earlier introduced the rocket equation, understood the potential of this finding on increasing the exhaust velocities. Tsiolkovsky's only flaw was that he was considering a 'flux of electrons' instead of ions, but the reason for this was that at that time the concept of atomic-sized particles having positive charge was not fully understood yet [10]. So for Tsiolkovsky, the flux of electrons is basically as close as he could have gotten at that time. Most Tsiolkovsky's work was theoretical so he never created any designs nor did he test the concept of electrical propulsion.

The other one who introduced the electric propulsion concept was the Robert Goddard, claimed to be one of the founding fathers of modern rockets (Together with Tsiolkovsky, Hermann Oberth, and Robert Esnault-Pelterie). Goddard visioned the potential of the electric propulsion already at 1906 and from there worked with it infrequently until in 1917 when he applied for a patent titled "Method of and Means for Producing Electrified Jets of Gas" [11]. The patent included drawings of three machines, including the one shown in figure 1, which can be stated as the worlds first documented electrostatic ion thruster [10]. At that time the United States also entered the First World War, which made Goddard focus almost entirely on chemical rockets the rest of his career.

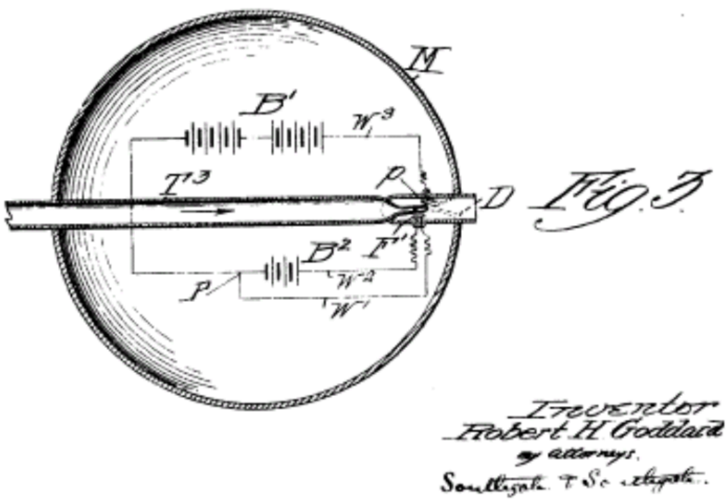

Figure 1 - Goddard's Ion thruster [10]

Similarly, to Tsiolkovsky and Goddard, Yuri Kondratyuk visioned electric propulsion potential and had a section for it in his manuscript, "To whomsoever will read in order to build", which was dated 1918-1919 [10]. In the same manuscript, he stated the potential of using solar energy to power the electric propulsion system. Also similarly, to Tsiolkovsky and Goddard, he prioritized chemical rockets over the electric propulsion [10] leaving the development to the future generations.

Hermann Oberth then continued this by presenting several different electric propulsion concepts in his groundbreaking book “Ways to Spaceflight” [10]. Still, it took 35 years more until Ernst Stuhlinger in 1964 described the first systematic analysis of electric propulsion systems in his book “Ion propulsion for space flight” [12]. The physics of an electric propulsion system was described by Robert Jahn in 1968 and a complete technology presentation was made by George Brewer in 1970. From there on the research moved to big laboratories like NASA’s Glenn Research center.

The first early stage electric thrusters were launched by U.S and USSR in the early 60s, they used Cesium and mercury as propellants [13]. USSR was able to make the first electric propulsion systems on an actual satellite when they used Hall thrusters to keep communication satellites in the right orbit [14]. The first ion thruster was launched by Japan in 1995 [15]. NASA made their first ion thruster launches with Hughes Aerospace’s Xenon Ion Propulsion System (XIPS) in 1997 [16] and a year later with NASA’s own NASA Solar Electric Propulsion Technology Applications Readiness (NSTAR) on the Deep Space 1 satellite [17]. European Space Agency (ESA) has also had a mission with ion thruster when they used Snecma’s Hall thruster on SMART-1 satellite [18].

During the last decades, the popularity of electric propulsion has increased rapidly and the trend seems to continue. The technology is already in the 4th generation and as the number of satellites using electric propulsion keeps increasing the amount of data received is also increasing. This helps to evolve the technology even further. It is predicted that during the 2020 half of the new satellites would be equipped with an electric propulsion system [19].

1.1.1 Electric thruster types

The most typical electric thrusters are described and divided by their main thrust producing technology in this chapter. The introduced thruster types are only the most common ones. There is a wide variety of different types of electronic propulsion devices, but they are far too numerous to be all introduced here. On figure 2 the division is visualized. The thruster type that was selected to be build while doing this study is highlighted.

We ended up selecting electrostatic thrusters because they are the furthest developed thruster types and currently also the most widely used. They are fairly simple to produce and they are proven to be reliable with high efficiency and specific impulse.

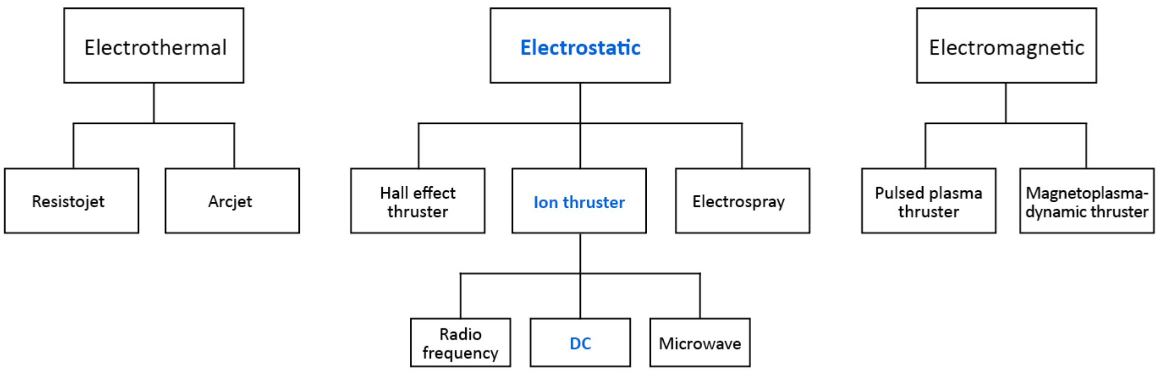

Figure 2 - Chart from different electric propulsion types

1.1.1.1 Electrothermal

The thruster is considered to be in the electrothermal group if the propellant is heated leading it to expand and then lead out from a funnel shaped nozzle, causing it to accelerate and give thrust on the spacecraft.

1.1.1.1.1 Resistojet

In resistojets propellant is heated in upstream in a resistively heated chamber or a similarly heated element and then lead to the nozzle. The velocity of the leaving particles is higher because of the thermal expansion and shape of the nozzle. This limits the specific impulse to below 500 seconds [13].

1.1.1.1.2 Arcjet

Arcjet heats the propellant by passing it through a high current arc. Because the propellant is ionized the effects on the exhaust velocity are insignificant even though it passes an electric discharge. The specific impulse limit is a bit higher than on the resistojet remaining below 700 seconds [13].

1.1.1.2 Electrostatic

When the particles are accelerated primarily by the Coulomb's force, which is a function of a static electric field (wherefrom the name) causing acceleration to the particles according to the Coulomb's Law, the thruster is categorized under electrostatic group.

1.1.1.2.1 Hall effect thruster

Together with ion thrusters Hall effect thrusters are one of the most used types of thrusters. Hall effect thrusters receive its propulsion thrust from generating plasma from a cross-field discharge known as the Hall effect. Hall effect thruster creates an electric field, which is perpendicular to a magnetic field. These fields electrostatically accelerate ions, which then exit from the downstream side of the thruster causing thrust to the device. The specific impulse and therefore efficiency is slightly lower when compared to ion thrusters, but the produced thrust is stronger and because of the simplicity of the device it is easier to build [13].

1.1.1.2.2 Ion thruster

There are many different techniques used in ion thrusters to generate plasma, which is used to ionize the propellant. Then by using electrically biased grids they electrostatically extract ions from the plasma and accelerate them to high exhaust velocities. The voltages can be more than 10kV. Ion thrusters have high efficiency and high specific impulse. They are also one of the most developed thruster techniques. Because of all the encouraging results received from this type of thrusters and also because of the manufacturing simplicity, this is the type of thruster that was selected for design in this thesis.

These thrusters could be further split by the plasma generation techniques (discussed more in detail later). The most common one and the one we decided to use, is the DC. There is also radio frequency and microwave plasma generators. The working principles of these are explained more in details on the Ion components chapter.

1.1.1.2.3 Electrospray and field emission electric propulsion thruster

Electrospray thrusters (or colloid thruster) generates the thrust by using small needles to where from it shoots electrostatically accelerated ions. It uses charged liquid as a propellant where it extracts the ions.

Field emission electric propulsion thrusters (FEEP) uses liquid metal as a propellant. It extracts ions by using the field emission process from the tip of a metal surface, which shoots the ions with high speed. Both of these thrusters provide very small thrust (less than 1 mN) and are therefore used mainly for precision control of spacecraft’s position or attitude control [13].

1.1.1.3 Electromagnetic

Thrusters using electromagnetic propulsion principle are using electromagnetic force known as a Lorentz force, which is a force perpendicular to the magnetic field and to the current in the device. This repulsing force is the one causing the propulsion.

1.1.1.3.1 Pulsed plasma thruster

Pulsed plasma thruster (PPT) (or plasma jet engine) is using a solid propellant, which is ionized with a pulsed discharge. By using a plasma arc part of the solid propellant is ablated, which then turns into gas because of the heat. The propellant is then accelerated by Lorentz force to exhaust with high velocity. The thrust level can be controlled by the frequency of the pulses.

1.1.1.3.2 Magnetoplasmadynamic thruster

Magnetoplasmadynamic thrusters ionize propellant with high efficiency by using very high current arc and then by Lorentz forces accelerates these ions. The advantage of this type of thrusters is that they are capable to use very high power, which also increases the produced thrust. In table 1 has been summarized the main properties of the thrusters presented above. The table was brought together by Martines-Sanches and Pollard [20] and it is already 19 years old, but still summarizes well the typical features of the different propulsion types.

Table 1 - Different thruster types and usual specifications

Thruster Type	Resistojet (N_2H_4)	Arcjet (N_2H_4)	Hall (Xe)	Ion (Xe)	FEEP (Cs)	PPT (Teflon)	MPD (self-field)
Power (W)	500 – 1 500	300 – 2 000	300 – 6 000	200 – 4 000	10^{-5} - 1	1 - 200	200 – 4 000 K
I_{sp} (s)	300	550	1 600	2 800	6 000	1 000	2 000 – 5 000
η (%)	80	35	50	65	80	7	30
Mass (kg/kW)	1 - 2	0.7	2 - 3	3 - 6		120	
Lifetime (h)	500	> 1000	> 7000	10 000		> 10^7 pulses	
S/C interaction concerns		Thermal radiation	Wide plume, ion backflow, torque operational	Ion backflow, non-propellant effluent	Cs contamination	Plume condensation	
Status	Operational	Operational	Operational	Operational	Development	Operational	Development
Typical mission	Orbit insertion, deorbit	Orbit transfer (medium ΔV)	NSSK orbit raising (medium ΔV)	NSSK orbit transfer (large ΔV)	Small orbit corrections	Small orbit corrections	Large ΔV
References	21, 22	23, 24	25, 26	27, 28, 29	30	31, 32	33

1.2 Equations and basic propulsion concepts

In this chapter, basic concepts and equations are presented. The formulas are mainly taken from the Goebel's and Katz's book fundamentals of electric propulsion [13] if not stated otherwise.

1.2.1 Specific impulse (I_{sp})

Specific impulse (I_{sp}) is the most common way to compare the efficiency of rocket and jet engines. The units typically are in seconds (s). Specific impulse is the ratio of thrust (T) and mass flow of propellant ($\dot{m}_p$) multiplied gravity acceleration (g), in the case of thrusters, thrust divided by mass flow rate can be changed to the exhaust velocity (v_{ex}) giving us the final form:

$$I_{sp} = \frac{T}{\dot{m}_p g} = \frac{v_{ex}}{g} \tag{1}$$

As we can see from the equation (1) above, a system, which has a higher specific impulse, gets more thrust out of the propellant. Electric propulsion devices typically have much larger specific impulse than chemical rockets. When chemical rockets can have a specific impulse of 200 to 500 seconds depending on the propellant used, ion thrusters can have a specific impulse exceeding 3 000 seconds. This is caused by the much higher exhaust velocity. Since electric propulsion devices have a smaller mass flow than chemical rockets the specific impulse is higher, but similarly, the thrust is also smaller. Because of the high specific impulse electric propulsion systems have better efficiency than chemical rockets. With this better efficiency, it is possible either to reduce the amount of propellant needed to pack with the spacecraft.

1.2.2 Other thruster efficiency measurement methods

This efficiency expresses how big portion of the propellant the thruster is able to take advantage by turning them into ions and accelerating them out, also known as the mass utilization efficiency. This is expressed in the equation (2) below

$$\eta_m = \alpha_m \frac{I_b M}{q \dot{m}_p} \tag{2}$$

Where α_m is mass utilization correction factor, I_b is the beam current, q is the charge and M are the ion mass. The way to calculate it is given below

$$\alpha_m = \frac{1+\frac{I^{++}}{2I^{+}}}{1+\frac{I^{++}}{I^{+}}} \tag{3}$$

Where I^+ is the singly charged ions and the I^{++} is the doubly charged ions. Another way to measure the efficiency of the thruster is known as the electrical efficiency (η_e) and this done by measuring the input power and then measure the power in the beam (P_b). This is given by

$$\eta_e = \frac{P_b}{P_T} = \frac{I_b V_b}{I_b V_b + P_0} \tag{4}$$

Where P_0 is the other power input required to make the thruster work (e.g. heating of cathode). P_b is the beam power and P_T is the total power going into the thruster. Ionization of the particles requires power as well, which is known as the discharge loss (η_d). It is a way to calculate how much power is lost in ion producing

$$\eta_d = \frac{P_d}{I_b} \tag{5}$$

where P_d is the power required to produce ions, the out coming units of this equation is electron-volts per ion (eV/ion), which are common units to compare the efficiency of different thrusters. Since it is a loss measurement, it should be as low as possible. When the discharge loss is high, it means that higher percentage of the propellants mass is being utilized and vice-versa. When the performance of a plasma generator is visualized it is done by plotting the discharge loss as a function of propellant utilization efficiency, as is shown in the figure (3)

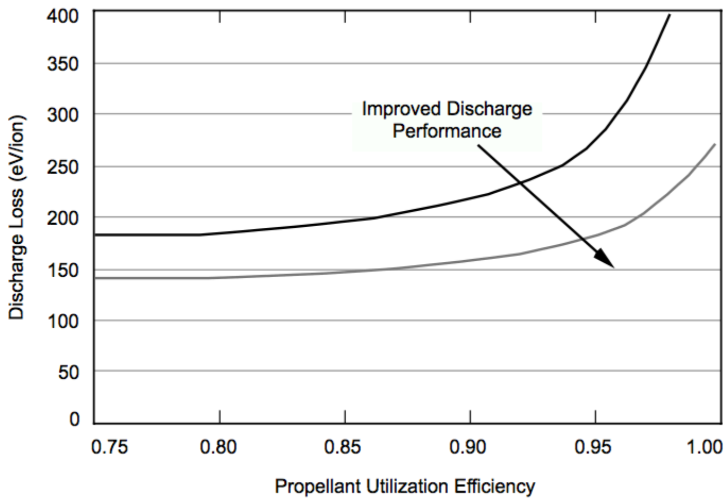

Figure 3 - Thruster Performance plotted with discharge loss against the propellant utilization efficiency [13]

Another way to measure the efficiency of a thruster is known as the total efficiency. This is the kinetic power produced by the thruster, also known as the jet power, divided by the total power input in the thruster.

$$\eta_T = \frac{T^2}{2\dot{m}_p P_{in}} \tag{6}$$

Where P_{in} is the input power. The formula to calculate this is shown in the equation (7).

$$P_{in} = \frac{I_b V_b}{\eta_e} \tag{7}$$

The total efficiency is commonly the referred efficiency when the efficiency of a thruster is discussed. From all the above formulas (including the ones in the specific impulse chapter) we can conclude that high exhaust velocities increase the specific impulse and therefore increases the efficiency. The specific impulse increases also when the mass of the ions is decreased, but to keep the desired thrust, power levels need to be increased accordingly. From this, we can conclude that with propulsion devices it is always a trade off between thrust and specific impulse, as already seen from the equation (1). The only way to avoid this trade off is to increase the input power, which seems to be the trend on the new ion thrusters.

1.2.3 Thrust

When an engine shoots the propellant out of the thruster it creates reaction force, known as thrust. Thrust is a combination of Newton's second and third law. While the spacecraft is using the thruster it is reducing its own mass because it is consuming propellant all the time, this is why the thrust is given by the time rate of change of the momentum, which is expressed in equation (8),

$$T = \frac{d}{dt}\left(m_p v_{ex}\right) = \frac{dm_p}{dt} v_{ex} = \dot{m}_p v_{ex} \tag{8}$$

Where T is the thrust. In the case of ion and Hall thrusters, the final expression can be translated to ion mass flow ($\dot{m}_i$) and ion velocity (v_i) when leaving the thruster. This is because the velocities of the leaving ions are far greater than the for ions that are not been ionized. Using the law of conservation of energy, ion velocity can be calculated with the equation (9)

$$v_i = \sqrt{\frac{2qV_b}{M}} \tag{9}$$

Where V_b is the net voltage. The ion mass flow rate ($\dot{m}_i$) is given by

$$\dot{m}_i = \frac{I_b M}{q} \tag{10}$$

When both of these equations are substituted into the equation (11) above we finally get the equation of the thrust for a single charged propellant,

$$T = \sqrt{\frac{2M}{q}} I_b \sqrt{V_b} \tag{11}$$

This is the simplified thrust equation and it works if the thrust is considered to be unidirectional, singly ionized and monoenergetic beam of ions. Typically, with ion thruster, there is a portion of ions that are multiply charged and a factor for the divergence of the beam needs to be added as well to the equation. In the case of ion thrusters, the monoenergetic ion beam assumption is typically valid.

Let's start from the charged ions. Usually, there is a certain amount of doubly charged ions. These ions have higher current. This changes the total beam current in the way as expressed in the equation (12)

$$I_b = I^{+} + I^{++} \tag{12}$$

There actually can be also ions that are charged more than two times as well, but usually, the number of these ions is so small that they can be neglected. To get the thrust corrected a factor is needed, which is a ratio of the doubly charged ions. This factor is expressed in the equation (13)

$$\alpha = \frac{I^{+} + \frac{1}{\sqrt{2}} I^{++}}{I^{+} + I^{++}} \tag{13}$$

The beam divergence is simple if the beam spreads uniformly from the thruster. It can be simply expressed as an effective thrust angle, which can be calculated with equation (14)

$$F_t = cos\theta \tag{14}$$

Where θ is the average half-angle divergence of the beam. If the spread is not uniform and it has some curves, it needs to be integrated, which is expressed in the equation (15)

$$F_t = \frac{\int_0^r 2\pi r \, I(r) \, cos\theta(r) dr}{I_b} \tag{15}$$

Where *I(r)* is the ion current density as a function of the radius. This is a value that typically is measured directly from the plume with a plasma probe. If the value of I(r) is constant equation (14) can be used for the calculation of the effective thrust angle (F_t). Together with these corrections the final form of the thrust equation can be achieved

$$T = \alpha F_t \sqrt{\frac{2M}{q}} I_b \sqrt{V_b} \tag{16}$$

1.2.4 Rocket equation

Tsiolkovsky rocket equation, also known as the ideal rocket equation, describes the motion of vehicles that follow the basic principle of a rocket. Rocket can be described as a device that applies acceleration to itself. The equation calculates the maximum change in the rockets speed (Δv) if there are no other external forces affecting it, by using exhaust velocity and initial and final masses.

$$\Delta v = v_{ex} \cdot \ln\left(\frac{m_i}{m_p}\right) \tag{17}$$

As Goebel and Katz state ‘high delta-v missions are often enabled by electric propulsion because it offers much higher exhaust velocities and Isp than conventional chemical propulsion systems’. While chemical rockets are restricted with exhaust velocities around 4 km/s, ion thrusters are able to reach up to 40 km/s exhaust velocities [13]. These high exhaust velocities benefits can be seen in the mass of used propellant, which is shown in the equation (18)

$$m_p = m_d \left(e^{\frac{\Delta v}{v_{ex}}} - 1\right) \tag{18}$$

where m_d is the final delivered mass and is given by

$$m_d = M_i \cdot e^{\frac{\Delta v}{v_{ex}}} \tag{20}$$

where M_i is the initial wet mass of spacecraft. From the equation (20) it is possible to see how the higher exhaust velocity has a dramatic effect on the propellant used. For example, a payload weighing 1 000 kg with a mission Δv of 3 km/s, a chemical rocket having an exhaust velocity of 3 km/s uses almost 1 720 kg of propellant while an electric propulsion thruster having an exhaust velocity of a 20 km/s uses 162 kg of propellant. Using electric propulsion systems enables to either reduce costs by the decreasing weight on the launch or to increase the dry weight of the spacecraft.

1.2.5 Delta-v

Delta-v is one of the most basic concepts in the aeronautics field. Simplified it is the change in velocity that is required to perform a manoeuvre. It is not the physical change in the vehicle's speed, but instead, it's a scalar vector, which has units in speed.

To calculate the delta-v you need to know the thrust on that specific time (T(t)) and mass at the specific time (m(t)). The first form can be simplified if there are no external forces with a coordinate acceleration ($\dot{v}$)

$$\Delta v = \int_{t_0}^{t_1} \frac{|T(t)|}{m(t)} dt = \int_{t_0}^{t_1} |\dot{v}| \qquad (21)$$

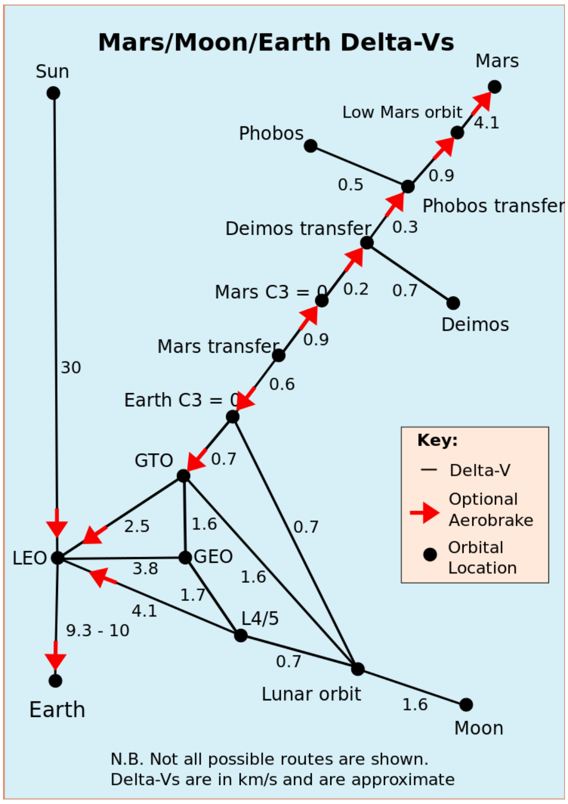

Figure 4 – Pork chop plot for different orbital transfers [34]

When talking about a mission's delta-v, we are typically talking about the required delta-v budget, which is an assessment of how much of delta-v is required for the whole mission. This is needed for the designers to know how much propellant is needed. The delta-v budget can be visualized with a "pork chop" plot, which is a diagram showing all the required changes in velocity to complete the manoeuvre. To illustrate this figure 4 shows a porkchop plot. The image is from Wikipedia from the delta-v article and the values are collected from an article on the internet [35] and by using a delta-v calculator on the internet [36], and is only for the sake of an example.

One of the most typical application using delta-v is the Hohmann transfer calculation, which calculates the required change in speed when an object is changing an orbit.

1.3 Ion thruster components

Gridded ion thruster works by bombing neutral propellant with electrons. These electrons are created in a hollow cathode by heating, which starts thermionic emission. The electrons are then shot into the discharge chamber where they fall towards the positively charged anode. While attracted towards the anode the electrons bombard the neutral propellant inside the discharge chamber and generate plasma. The generated plasma is drawn by the grids and accelerated to high exhaust velocities (exceeding 30 km/s). A neutralizer cathode, which emits electrons at the same rate as the ions exit from the thruster, is often used to avoid that the thruster does not have an increasing negative charge, which would cause the exiting ions to be attracted back to the thruster and therefore decrease the thrust. On figure 5 the thruster working principles are visualized.

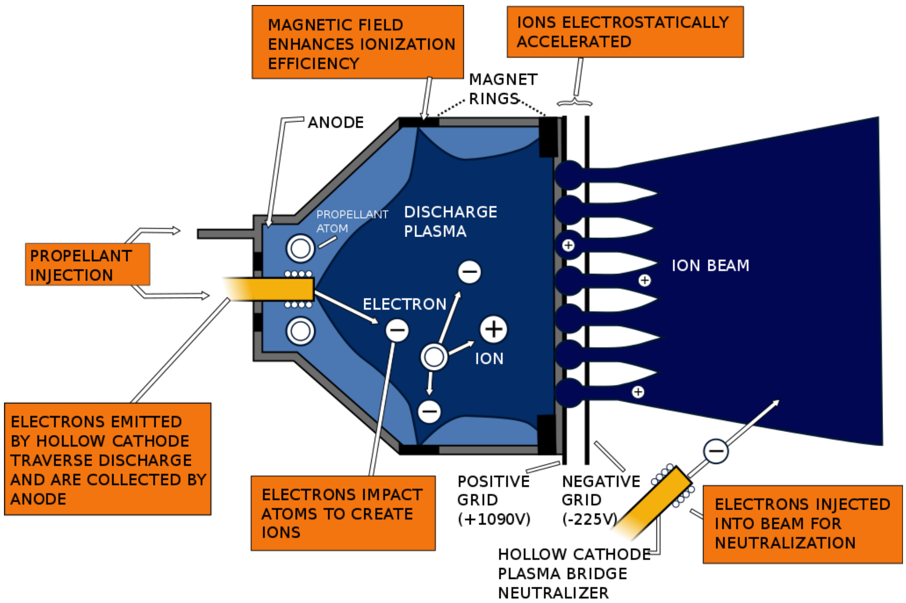

Figure 5 - Thruster working principle [37]

1.3.1 Ion Accelerator

Ion accelerator (also known as ion optics), is the grid system that draws ions from the discharge chamber and accelerate them to high exhaust velocities. It consists of screen grid, accelerator grid and sometimes from deceleration grid. Each grid has its own job. Screen grid is positively charged and extract ions from the discharge chamber and protects accelerator grid. Second grid is the accelerator grid, which accelerates the ions and the last one deceleration grid prevents electrons from back flowing [13]. When the grids are designed the designer needs to choose does he emphasise performance, life or size. Since most of the ion thrusters are supposed to work for several years', life is commonly the most important factor. An ideal grid would drag all the approaching ions while same time reflecting the neutral gas, then accelerate these ions creating a thrust without having any erosion and keeping the power and thermal levels all the time constant.

When the particles get closer to the grids, there is potential and density variations in the plasma to satisfy the particle balance or the imposed electrical conditions from the discharge chamber walls. These regions with the variations are known as sheaths. One of the most typical sheaths occurs when potential is very large compared to the electron temperature [13], this type of sheath is known as the Child-Langmuir sheath. In equation (22) is expressed the most common form of the Child-

Langmuir law, where J_i is the current density, ε_0 is the permittivity of free space and d is the distance between the electrodes. The distance between the electrodes, in case of ion thrusters, is the diameter of the accelerator grid, this is also known as the Child-Langmuir length [13].

$$J_i = \frac{4\varepsilon_0}{9}\left(\frac{2q}{M}\right)^{1/2}\frac{V^{3/2}}{d^2} \tag{22}$$

To accelerate the ions to high energy the plasma boundary distance should be around Child-Langmuir length so that a sheath can be formed, which will accelerate the ions and reflect electrons. Since the Child-Langmuir length is proportional to the mass of the ions the grids need to be optimized for the used propellant [13]. From the Figure 6, we can see that the aperture size for different propellants. As the atomic mass of the propellant increases the Child-Langmuir length decreases accordingly. Also, increasing the ion current decreases Child-Langmuir length.

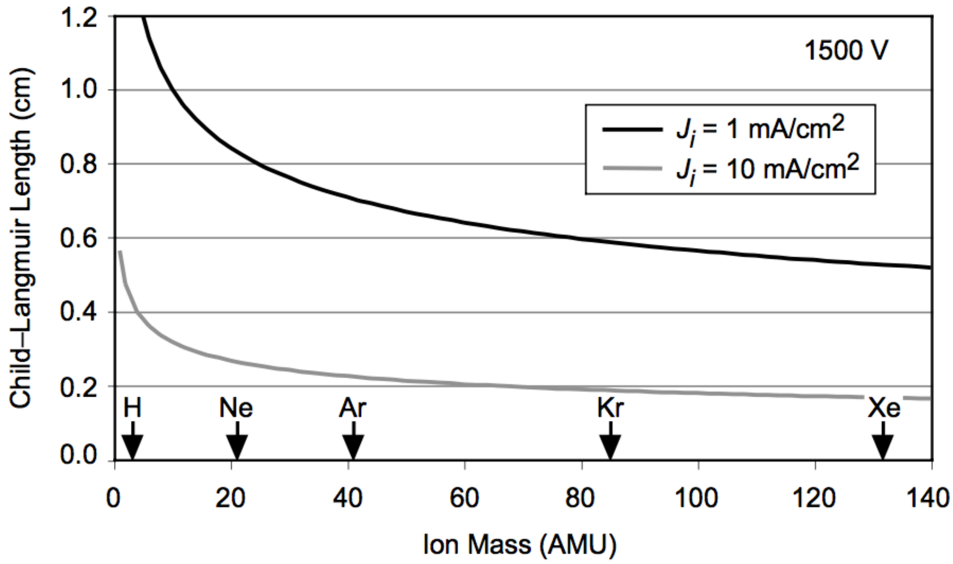

Figure 6 - Child-Langmuir Sheath length against Ion Mass [13]

Ions going through the aperture of the accelerator grid are adding up to form the beam, while ions that miss the aperture hit on the face of the accelerator grid and causing it to erode eventually. This is why a “screen” grid is used to “guide” the ions so that they can’t hit the accelerator grid. The screen grid is typically either floating electrically or is biased to the cathode so that it reflects electrons back into the plasma causing the ions hitting on the screen to have low energy [8].

On figure 7 we can see grid settings from a few well know gridded ion thrusters. The comparison of settings includes the distance between the two grids apertures from center-to-center (l_{cc}), accelerator grids hole diameter (d_a), screen grid hole diameter (d_s), the thickness of accelerator grid (t_a) and screen grid (t_s) and the space between grids (l_g) [8].

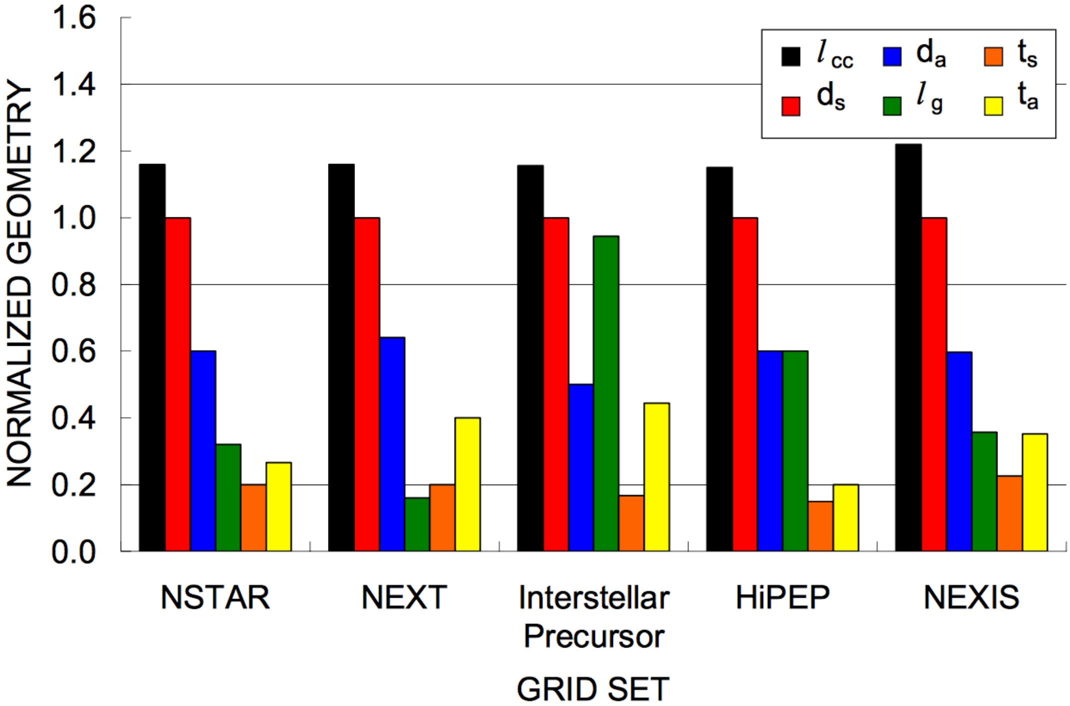

Figure 7 – Comparison of grid settings of five ion thruster [8]

1.3.2 Plasma generator

There are three different types of plasma generators; Direct current electron (DC) discharge, radio frequency (RF) discharge and microwave (MW) discharge. In here I'm going to represent how a DC electron discharge plasma generator works. RF and MW plasma generators work almost identically and they are explained at the end of this chapter.

DC electron discharge unit consists of an anode potential discharge chamber with a hollow cathode (explained in more detail in its own chapter), which works as a source for electrons. With these components, plasma is generated into the discharge chamber and from this plasma, the ions are extracted and accelerated to form the thrust beam. The neutral propellant is added to the discharge chamber as well to the hollow cathode. The propellant gas is then ionized with the electrons that are obtained from the hollow cathode.

The efficiency of the discharge process can be improved by applying sort of a magnetic "pusher" to the wall of the anode. The magnetic fields prevent the electrons from hitting the anode walls and this way increases the electrons flying paths length, which again increases the probability for the ionization of the neutral gas. It is critical to have just the right strength on the magnetic field to ensure that the electrons get trapped in it, but without causing too high electron loss on the anode, making sure that the discharge remains stable throughout the whole operating range of the thruster.

The RF and MW ion thrusters use almost similar design with the ion accelerator and with the electron-neutralizer as the DC discharge thrusters. The difference with them is that instead of having a hollow cathode or an anode power supply they have an antenna structure, which is used to ionize the propellant gas. Similarly, to the DC discharge thrusters, these thrusters also uses self-generated magnetic fields to improve the ionization efficiency.

1.3.2.1 Magnetic fields

There have been ongoing studies of magnetic fields in the discharge chamber and through these studies, ring-cusp fields have shown most potential [13]. The magnetic fields need to be just right for them to increase the thruster efficiency. Typically, this is done by simulating the whole discharge chamber with the magnetic fields. There is plenty of studies done from this. Goebel [13] for example has created a model known as 0-dimensional (0-D) model. The model calculates the main features inside the discharge chamber including creating a detailed model of the magnetic field [13].

1.3.3 Hollow Cathode

Hollow cathode is the hardest part to manufacture in ion thruster. Hollow cathode produces electrons, which first ionizes the propellant and then neutralize the exhaust propellant. The manufacturing of the hollow cathode is hard because it is a very precise and compact piece of equipment, which needs to be heated to high temperatures. The heating is required for the cathode to start emitting electrodes which then starts the whole event chain ending in thrust. Typical cathode configuration can be seen from the figure 8.

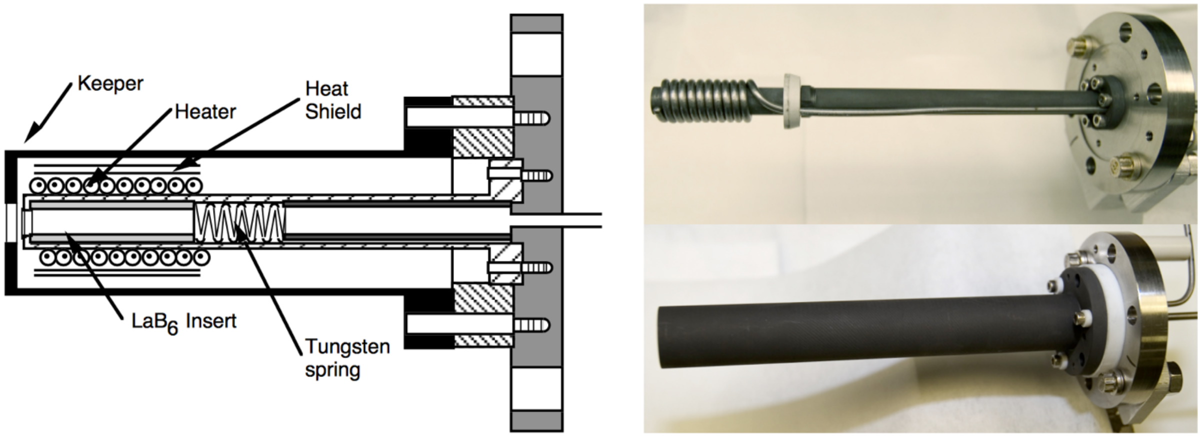

Figure 8 - Typical cathode construction [38]

The temperature required by the hollow cathode depends on the insert material used. First ion thruster models suffered from using of tungsten filaments, which were required to be heated around 2 300 °C to start the ionization process [13]. This caused problems with heat control and lowered the efficiency of the thrusters. The solution for this was to use hollow cathodes where an insert is set inside a tungsten pipe. The insert starts the thermic emission lower temperatures making the thermal design easier. The type of the insert depends on the mission. At NASA there is an ongoing research for finding more efficient insert materials without losing the lifetime [38]. NASA has been using barium-oxide impregnated tungsten hollow cathodes (BaO-W). They have a work function of 2.06 eV, which is just over 1 000 °C. Because producing the cathodes is a chemical process the BaO cathodes are easily poisoned and therefore require very pure propellants, for this NASA has created "propulsion-grade" xenon with >99.999 percent purity [38]. Also, NASA is in need of increasing the discharge current received from the cathode, which would cause overheating of the BaO cathode and shorter lifetime of the thruster. In the figure, 9 has presented different insert materials and their thermic emissions as a function of temperature.

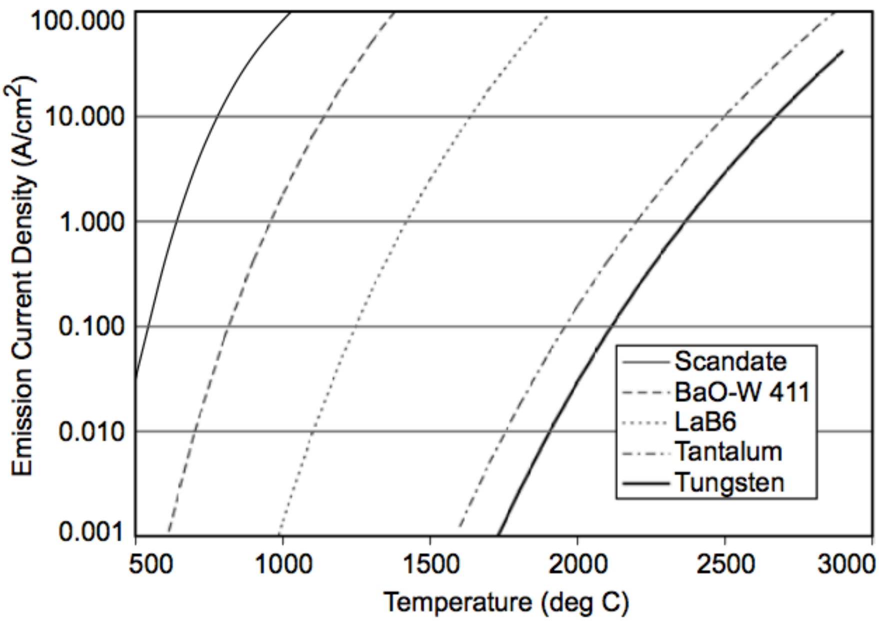

Figure 9 - Thermic emission current rate as a function of temperature [13]

From the figure 9, it is easy to see why the use of tantalum and tungsten cathodes has been discontinued. BaO and scandate cathodes seem most interesting from the work function point of view. The downsides of the BaO has explained earlier already. The scandate results presented here is received by adding scandate to the surface of a BaO cathode so it has the same downsides as the BaO, although the emission rates are much higher than pure BaO [13]. NASA has been researching lanthanum hexaboride (LaB6) already from the 1970s [13], which has a work function at around 1 700 °C. Although the required heating temperature is higher, LaB6 have been very reliable and doesn't require as pure propellants either as BaO. The LaB6 hollow cathodes have been showing good lifetime and NASA has been able to receive 250 A discharge values with LaB6 hollow cathodes, which is ten times more than BaO cathodes are able to withstand. The drawback in LaB6 is that it evaporates boron on the supporting material. If the supportive material is made from a rigid material like tungsten it can cause a structural failure [38].

The amount of discharge can be estimated with Richardson-Dushman equation [13]

$$J = AT^2 e^{-\frac{q\phi}{kT}} \tag{23}$$

Where A is a universal constant, T is temperature and ϕ is the work function of the insert material. In addition of these, the insert's surface area should be added there. The work function as well the value for constant was received from Goebel and Katz [13].

1.3.4 Propellant feeding system

Propellant feeding systems are typically simple systems where you can control the propellant flow with a valve. Different propellants might have different requirements, for example, adamantane might clog the feeding system if the temperature drops inside the feeding system. To avoid this typically a modified system is built where it is possible to control the temperatures inside the system. Since the feeding systems are out of the scope of this study they are not covered more in this study.

1.4 Propellants

On an ideal propellant, the ionization threshold would be low and the cross-section area of the ionization would be high so that the propellant would be energy efficient. The low boiling point makes the thermal control easier and if availability is high the propellant is cheap, making it more attractable. Mass actually is not a very important feature of a propellant, because the force is made with electrostatic forces between ion charge and the extraction grid. Nevertheless, high mass improves the efficiency of the thruster by lowering the required power for the ionization and enabling higher thrust levels in lower power as already discussed earlier [39].

Xenon at the moment is by far the most common propellant. It doesn't really shine on any of the requirements stated above, instead, it is the best compromise. Xenon is in a gaseous state at low temperature, it has relatively high molar mass (improves efficiency) and it is a noble gas so it doesn't react with the surface of the spacecraft. On other note, the ionization threshold is quite high and it is expensive (which is the main reason for searching alternative propellant) and the price seems to keep increasing [39].

All the key parameters of the presented propellants are added to table 2 for practicality. Xenon was also added there for comparison.

Table 2 - Key parameters of selected propellants

Propellant		Argon	Iodine	Adamantane	Diamantane	Krypton	Xenon
Ionization (kJ/mol)	1st	1 521	1 008	890.6	849.1	1 351	1 170
	2nd	2 666	1 846	1 360		2 350	2 046
	3rd	3 931	3 180			3 565	3 099
Molar mass (g/mol)		.95	126.9	136.2	188.3	83.8	131.3
Melting point (°C)		-189	113.7	270	245	-157.37	-117.8
Price (€/kg)		10	161.5	270			1 300
Reference		40	40	40, 41	40	40	40

1.4.1 Argon (Ar)

Argon is a noble gas and third common gas in the Earth's atmosphere. Like Xenon it is an inert gas, so it doesn't cause unwanted reactions with the spacecraft. Argon by far the cheapest propellant available. Argon also has the highest exhaust velocities, due to low weight. Argon is also flight tested and it has been used as a propellant for an ion thruster before.

Unfortunately, that's all the good things in argon, the molecular mass is less than one-third of the molecular mass of xenon and the ionization threshold is highest from the propellants selected in this study. The lighter molecular mass can be somewhat dealt with few different solutions. Firstly, the thruster could increase power, which would increase the thrust accordingly. Secondly, the mission is simply just planned the way that it takes longer for the satellite to reach the destination, for commercial satellites this is usually not suitable, as they make non-operational losses, but for scientific satellites, the longer orbit transfer might be negligible and there would be considerable savings in the fuel costs. Finally, at the higher specific impulse rates, the efficiency of the argon reaches almost on the same level as xenon.

1.4.2 Iodine (I)

Iodine offers, similarly to xenon gas, a high density and low-pressure storage. Iodine is an interesting propellant also because it is solid at room temperature. This is important especially for thrusters used in CubeSats as they are typically launched as a secondary payload and high-pressure systems often not allowed by the primary payload launch customer [42], it also consumes less volume when solid. Iodine is also fairly cheap (we were able to buy it for around 160 €/kg) at least when compared to xenon.

Molecular weight and ionization threshold is close to xenon's, but the cross-section area at higher energy levels is much higher than xenon's, making it very interesting propellant option for xenon. The negative feature of iodine is that the gas is corrosive, making the lifetime of a thruster or in the worst case whole satellite shorter [39]. Iodine is currently one of the most studied alternative propellants.

1.4.3 Diamondoids

Diamondoids are a relatively new introduction for propelling ion thrusters. Diamondoids are sp3-hybridized carbon clusters and they have a similar structure as cubic diamond lattice fully saturated with hydrogens [39]. Diamondoids are classified by the number of diamond cages. All the diamondoids have a low ionization threshold when compared to xenon. The double ionization threshold is not found from the literature, but Goulart et al. [41] measured it to be 1 360 kJ/mol. For diamantane, no value was found from literature.

Adamantane is a cycloalkane and the simplest and smallest of the diamondoids. Adamantane has many good features as an electric propulsion propellant, first, it starts evaporating already at room temperature [43], secondly it is quite cheap as it is a by-product of petroleum industry and finally it has low ionization level [39] (although, the adamantane founded for this study was significantly more expensive being 270€/kg).

Diamantane is the second-smallest of the diamondoids. It has the largest molecular mass of the propellants presented here and lowest ionization threshold. It starts evaporating at 62 °C degrees [43]. The price of diamantane is higher, then on adamantane, but we were unable to confirm this because our supplier didn't have it.

All the diamondoids are negatively attracted to electrons, which lowers the propellant efficiency. It is possible that it can be solved by modifying the molecular structure of diamondoids and this way make them more suitable as propellants [44-46].

1.4.4 Krypton (Kr)

Krypton is, like argon and xenon, a noble gas and therefore also an inert gas. Krypton is slightly lighter than xenon, has slightly higher ionization threshold and it has a lower cost than xenon, but it is approximately 100 times more expensive than argon. Krypton has the lowest erosion rate at high-energy rates and it provides highest specific impulse [46]. Krypton has never been tested in space. Our supplier didn't have krypton so it was not selected for testing.

2 Design for the thruster

The goal of the thruster design was to create a working model, which would enable testing of alternative propellants and enable the use of 3D printing. The plan also was to keep costs as low as possible to make the concept more feasible for small satellite projects. A typical thruster design project would start from the mission. The thruster itself would be constrained by the available power and space on the spacecraft. Typically, these parameters already give a good scope of what kind of thruster is required.

In our case there basically was no limitations, the thruster was intended for laboratory use, so no power limitations since we could use the mains electricity and we only needed to be able to fit the thruster into the vacuum chamber (which actually constrained the diameter below 15 cm). The only real constraints were time, money and availability of the parts. Since we were planning to print the discharge chamber, we could use standard parts available from magnets and other parts. This way we were able to reduce costs even further. One of the desired features was being able to modify the test setup easily. This meant that all the parts should be easily switchable.

2.1 Discharge chamber

On the discharge chamber design, we hoped we could utilise 3D printing enabling a simplified and more unified build. The length and the diameter of the thruster were intended to keep small making it more affordable, but still enabling full testing capabilities. On the design, we decided to keep the length/diameter ratio close to 1 [6]. Typically, ion thrusters are built by using flanges for each part. These are then connected to each other by using rods. This is visualised in figure 10.

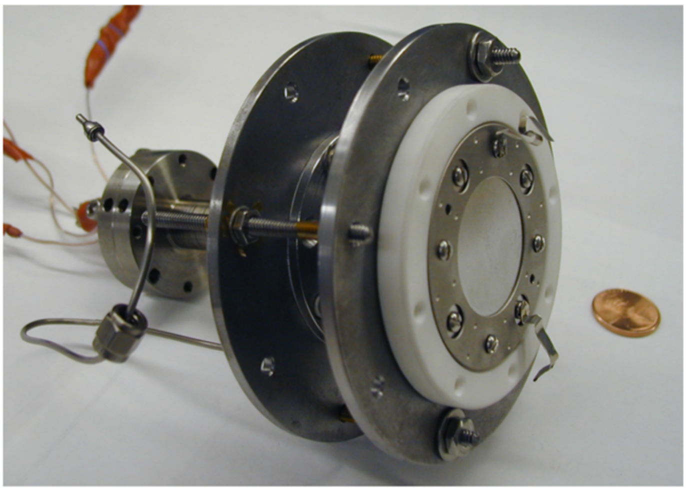

Figure 10 - Micro-ion thruster assembly [6]

In the effort of trying to simplify this design, we created an unibody design by benefiting 3D printing. The advantages of the unibody form are very easy manufacturing (3D print), simplifies and makes the body more compact. Cons of the design are size limitation (depends on the chamber where the 3D print is made) and material options are quite limited. The use of 3D printing technology allowed us arranging the magnets in whatever order we wanted. On a micro ion thruster, the difference of putting magnets on the outer surface or on the inner surface of the discharge chamber is negligible [6]. To simplify the design and make it easy to modify the settings of the magnets they were left outside. The magnets close to the grid are either disc or block magnets and the one around the cathode is a ring magnet. According to the Wirz et al. [6], the double ring cusp setting with a ring magnet around cathode leads to most efficient thrust.

The challenging part of having an unibody model would be on electric insulation. In our design, we were planning on using the body itself as an anode and then just insulating it from all the other parts. The insulation needing parts would only be the magnets and hollow cathode. Screen grid typically has the same voltage as the anode so no insulation is needed between it and anode [8]. The final design of the discharge chamber can be seen from the figure 11.

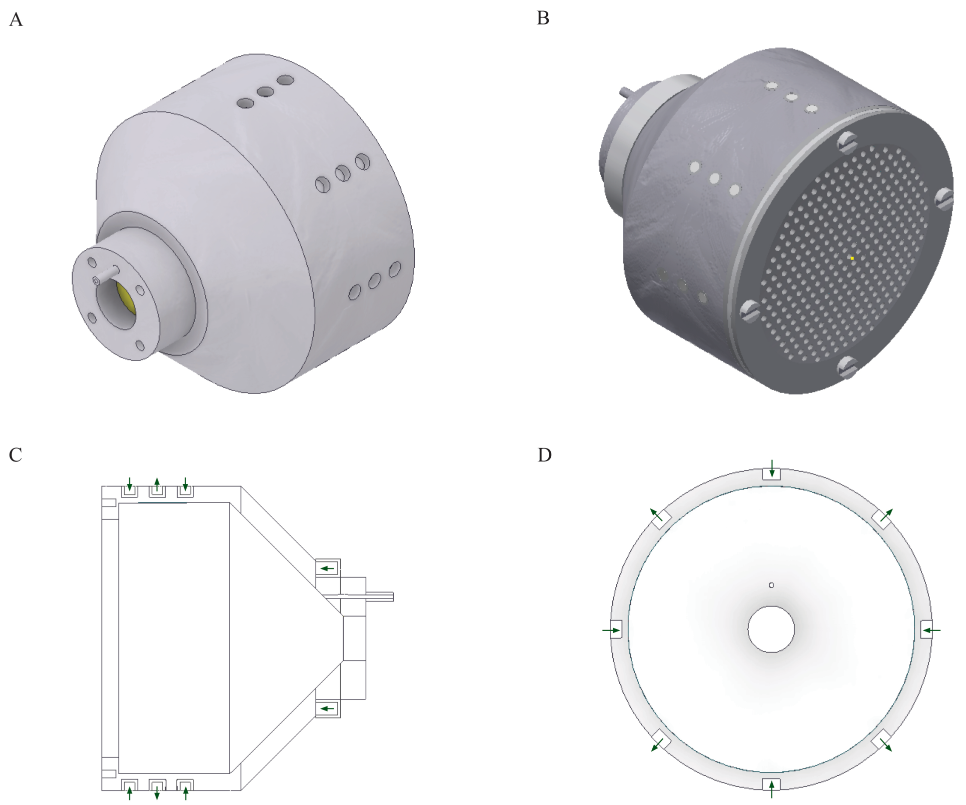

Figure 11 - Visualization of the discharge chamber. The pictures, C and D visualises the configuration of the magnets

2.2 Ion accelerator

Ion optics have a simple job. They extract ions from the plasma and then accelerate it to high exhaust velocities. Screen grids potential is smaller than discharge chamber's which is why it attracts ions from the plasma. Since our thruster was going to be only used in laboratory no decelerator grid was needed. If the accelerator grids evaporation would start to affect to thruster's performance the grid could be simply changed. Propellants might set own requirements for the grid material also. Iodine, which is corrosive can oxidise metal surfaces and therefore shorten the lifetime of the thruster. To avoid this different materials or coatings can be used. Another serious threat for ion optics is heat expansion. The thermal expansion is proportional to the diameter of the grids. Large ion thrusters use graphite or molybdenum grids to avoid this issue. On our scale, thermal expansion should be negligible. The hexagonal order for apertures was used to receive a high transparency to the ions. The grids are curved to make the beam shape more unidirectional.

The aperture size of the accelerator grid is minimized to maximise the grid transparency and for preventing neutral gas from leaving the discharge chamber. Normally, grids are made as thin as possible to save weight, but since weight was not an issue for us, the grids could be made thicker. This was required because we used 3D printing for the grids as well, and it was not possible to make the grids thinner than 1.2 mm with this technique. The aperture size for the screen grid was designed to be 2 mm and 1.2 mm for the accelerator grid. The ratio between screen grid and accelerator grid as seen already in figure 7 is 0.6. The grid material is stainless steel, mainly because it was the strongest material available from our 3D print supplier. The gap between the grids was 1 mm and it was done by 3D printing ABS plastic ring between the grids. The plastic ring worked as an insulator also. If the thermal stress would be too high for the insulator a ceramic spacer could also be used, made from aluminium oxide (Al_2O_3). The final cathode design can be seen from figure 12 on the next page.

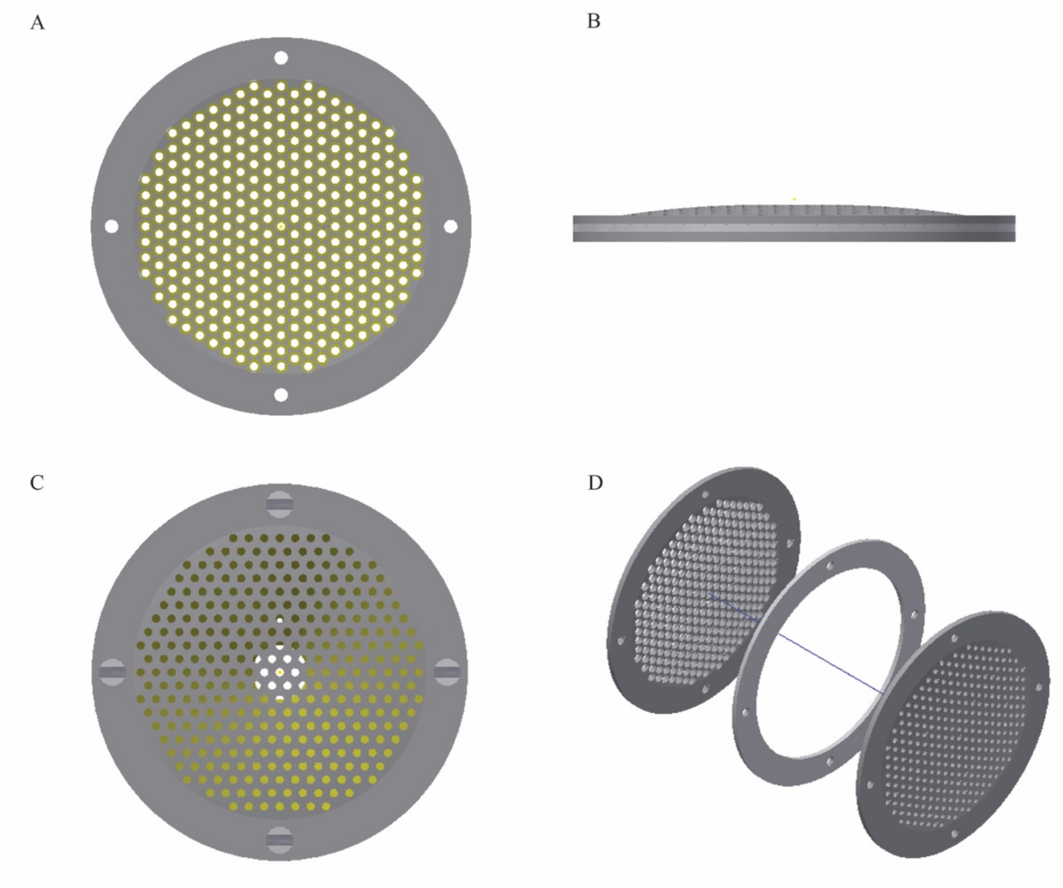

Figure 12 – Ion optics visualised

2.3 Hollow cathode

As already mentioned at the hollow cathode chapter (1.3.3) the manufacturing of hollow cathode is probably most difficult of any of the parts in ion thruster. Hollow cathode design should be started from the required discharge. This gives an inspection for the size of the cathode as well for the material. If discharge requirements are low a tungsten filament might provide enough thermic emission to ionize the gas, if the discharge requirements increase a hollow cathode is required. In our case, a hollow cathode was decided to be a more coherent solution as it would be more flexible in terms of the discharge current.

The insert was designed to have a 2 mm inner diameter and it is 2 cm long. The insert material is lanthanum hexaboride and with 1 300 °C heating it should give 291 mA of the discharge current when using equation (23). The heater is tantalum wire coated with alumina. Insulators are also made of alumina. The cathode tube has a 5 mm outer diameter and the keeper has 9 mm outer diameter. Both of these is made from poco (company) graphite. The heat shield is made of tantalum. The distance from the orifice of the keeper was designed to be close to the screen grid because according to Wirz et al [48] this increases the propellant efficiency and gives flatter plume. The design received plenty of inspiration from Goebel and Chu [38]. Visualisation of the design with an exploded model is shown in figure 13.

A B

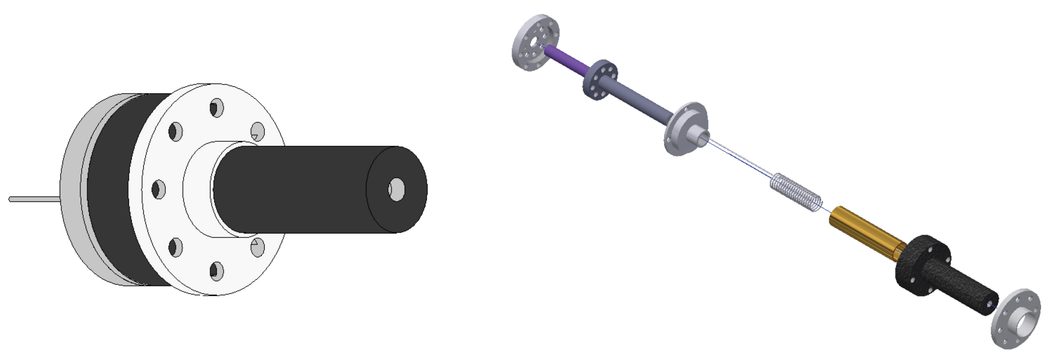

Figure 13 - A) The hollow cathode design B) Exploded view from the hollow cathode

2.4 Propellants

The featured design was made in mind that alternative propellants, like iodine, adamantane, and diamantane could be tested with the thruster. These propellants do not really require large modifications on the design but for example, the hollow cathode was considered as a better option because of these requirements. Since all the alternative propellants are in solid form at the room temperature the gas feeding system should be designed so that it enables altering of the storage temperature as well the temperature during the whole gas feeding process. The corrosive nature of iodine and its effects on the thruster should be monitored when testing is done.

2.5 Final design

The final design is shown in figure 15. The outside diameter is 54,8 mm and is made from aluminium. The full length of the whole thruster with all the parts assembled is 63,6 mm. The length of the discharge chamber is 48,6 mm. All the insulators are designed to be alumina, except the one between the grids, which is ABS. The magnetic cusps are implemented with 2 mm diameter disc magnets and they are insulated from the body with alumina cups. There is in total 8 magnets per row and there are 3 rows concluding in total 24 magnets. The back stream end of the magnet is 22 mm diameter and has a ring magnet with an alumina insulator. Everything is assembled with alumina screws to ensure that there are no short circuit problems. The electrical wiring is simple and it has been illustrated in figure 14 and it was taken from Wirz et al [5].

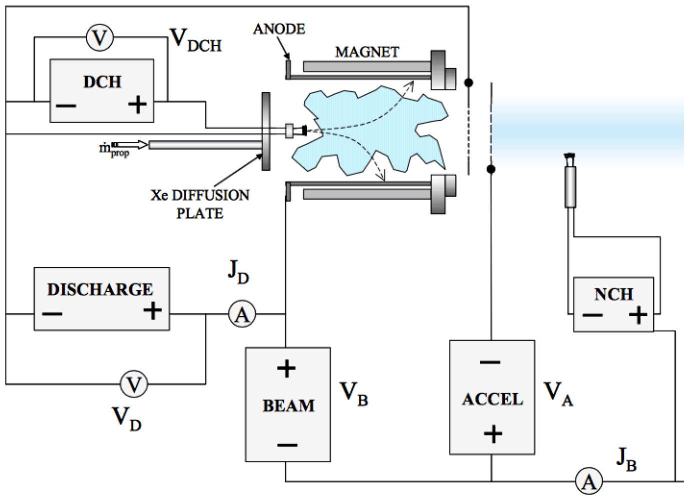

Figure 14 - Typical electrical configuration of a thruster [5]

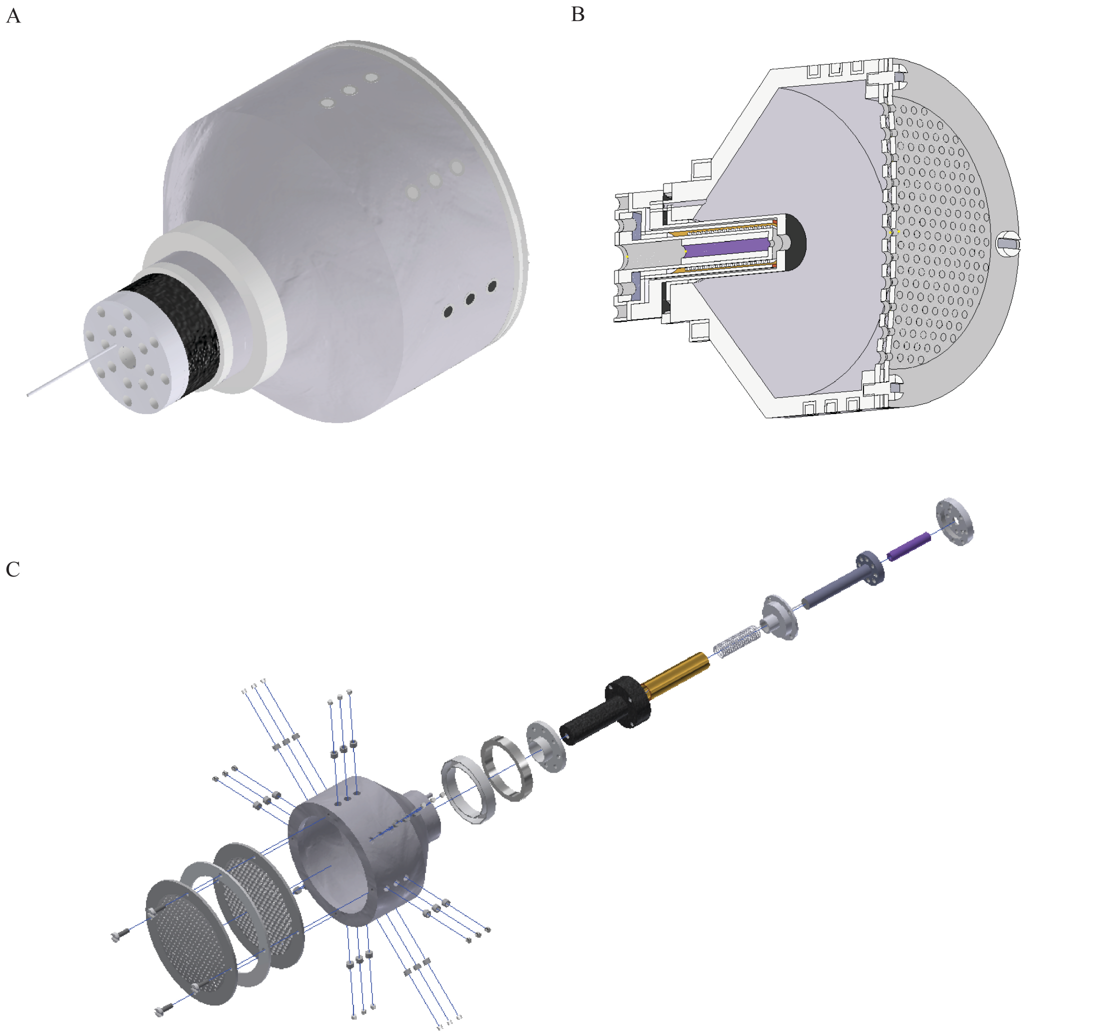

Figure 15 - The final design assembly. A) A view of the upstream part of the thruster
B) A half cut view from the thruster. C) Explode view from all the parts in the thruster

3 Conclusion and future studies

The original goal of this thesis was to create a robust design of a gridded ion thruster using DC discharge plasma generation technique. The goal was to design the system also so that it would make testing of alternative propellants possible as the price of xenon is high. Into the original goal was included building and testing as well. Considering authors low experience of the topic and the rigid schedule of the thesis, it was known already at the beginning that time considered to finish the work is short. As it turned out that was the right interpretation, and one month before the deadline the thesis was required to be shortened to be only a design of a gridded ion thruster.

The thruster was designed to have a 2 mN of thrust. The propellant flow and everything else was designed around this value. Voltage around the thruster and propellant flow needs to be adjusted accordingly to the used propellant.

The natural next step would be building the thruster and testing its performance. Also, a performance model with detailed simulation would be needed to make adjustments on design details like voltages around the thruster, screen and accelerator grids aperture sizes, the strength of magnets and hollow cathode design. Especially with the cathode author feels there would be room for improvement. Also, the propellants might need work before their potential can be fully utilized. Especially diamondoids look very promising on the paper but would require testing and maybe modification on the molecular structure, which would be as a study already in a magnitude of a full thesis.

Bibliography

1. Nasa.gov. (2017). Thruster Achieves World-Record 5+Years of Operation. [online] Available at: https://www.nasa.gov/centers/glenn/news/pressrel/2013/13-021_thruster.html - .WZgVqtOGMUE [Accessed 10 Aug. 2017].

2. Mueller, J., Hofer, R., Parker, M., and Ziemer, J. (2010). Survey of propulsion options for CubeSats. Pasadena, CA: California Institute of Technology.

3. W. Andrew Hoskins et al. 30 Years of Electric Propulsion Flight Experience at Aerojet Rocketdyne, Paper IEPC-2013-439, 33rd International Electric Propulsion Conference, Washington DC, October 2013.

4. Nasa.gov. (2017). Ion Propulsion: Farther, Faster, Cheaper. [online] Available at: https://www.nasa.gov/centers/glenn/technology/Ion_Propulsion1.html [Accessed 10 Aug. 2017].

5. Wirz, R. (2015). Miniature Ion Thrusters: A Review of Modern Technologies and Mission Capabilities.

6. Wirz, R., Polk, J., Marrese, C., Mueller, J., Escobedo, J. and Sheehan, P. (2001). Development and Testing of a 3cm Electron Bombardment Micro-Ion Thruster. [online] California Institute of Technology.

7. Boeva, J. (2014). A continuous plasma thruster using water vapour as propellant for nanosatellite propulsion. Delft University of Technology.

8. Farnell, C. (2007). Performance and lifetime simulation of ion thruster optics. [ebook] Fort Collins, Colorado: Colorado State University.

9. Mel'kumov, T. M., (ed.), Pioneers of Rocket Technology, Selected Works, Inst. for the History of Natural Science and Technology, Academy of Sciences of the USSR, Moscow, 1964; translated from the 1964 Russian text by NASA as NASA TT F-9285, 1965.

10. Choueiri, E. Y., A Critical History of Electric Propulsion: The First 50 Years (1906-1956), Journal of Propulsion and Power, vol. 20, pp. 193– 203, 2004.

11. Goddard, R.H., Method, and means for producing electrified jets of gas. US Patent No. 1,163,037. Application filed October 1917, granted December 1920.

12. Stuhlinger, E., Ion Propulsion for Space Flight, New York: McGraw-Hill, 1964.

13. Goebel, D. M, and Katz, I., Fundamentals of Electric Propulsion. 1st ed. Hoboken: John Wiley, 2008. Print.

14. Boever, A. S., Kiim, V., Koroteev, A.S., Latyshev, L.A., Morozov, A.I., Popov, G.A., Rylov, Y.P., and Zhurin, V.V., State of the Works of Electrical Thrusters in the USSR, IEPC-91-003, 22nd International Electric Propulsion Conference, Viareggio, Italy, 1991.

15. Shimada, S., Sato, K., and Takegahara, H., 20-mN Class Xenon Ion Thruster for ETS-VI, AIAA-1987-1029, 19th International Electric Propulsion Conference, Colorado Springs, Colorado, May 11–13, 1987.

16. Beattie, J. R., XIPS Keeps Satellites on Track, The Industrial Physicist, June 1998.

17. Brophy, J. R., NASA's Deep Space 1 Ion Engine, Review Scientific Instruments, vol. 73, pp. 1071–1078, 2002.

18. Koppel, C. R., and Estublier, D., The SMART-1 Hall Effect Thruster around the Moon: In Flight Experience, 29th International Electric Propulsion Conference, IEPC-2005-119, Princeton, New Jersey, October 31–November 4, 2005.

19. Forrester, C., Beyond Frontiers, Broadgate Publications (September 2016) pp 20

20. Martinez-Sanchez, M. and Pollard, J. (1998). Spacecraft Electric Propulsion-An Overview. Journal of Propulsion and Power, 14(5), pp.688-699.

21. Dressler, G. A., Morningstar, R. E., Sackheim, R. L., Fritz, D. E., and Kelso, R., Flight Qualiécation of the Augmented Hydrazine Thruster, AIAA Paper 81-1410, July 1981.

22. Miyake, C. I., and McKevitt, F. X., Performance Characterization Tests of a 1 kW Resistoject Using Hydrogen, Nitrogen, and Ammonia as Propellants, AIAA Paper 85-1159, July 1985.

23. Smith, W. W., Smith, R. D., Yano, S. E., Davies, K., and Lichtin, D., Low Power Hydrazine Arcjet Flight Qualiécation, Proceedings of the 22 International EP Conference (Viareggio, Italy), Centrospazio, Pisa, Italy, 1991 (IEPC Paper 91-148).

24. Birkan, M. A., and Myers, R. M. Introduction to Arcjets and Arc Heaters: Research Status and Needs Special Section, Journal of Propulsion and Power, Vol. 12, No. 6, 1996, p. 1010.

25. Kim, V., Main Physical Features and Processes Determining the Performance of Stationary Plasma Thrusters, Journal of Propulsion and Power, Vol. 14, No. 5, 1998, pp. 736 – 743.

26. Sankovic, J. M., Haag, T. W., and Manzella, D. H., Operating Characteristics of the Russian D-55 Thruster with Anode Layer, AIAA Paper 94-3011, June 1994; also NASA TM 106610, June 1994.

27. Kerslake, W. R., and Ignaczak, L. R., Development and Flight History of SERT II Spacecraft, AIAA Paper 92-3516, July 1992.

28. Beattie, J. R., Matossian, J. N., and Robson, R. R., Status of Xenon Ion Propulsion Technology, Journal of Propulsion and Power, Vol. 6, No. 2, 1990, pp. 145 – 150.

29. Sovey, J., et al., Development of an Ion Thruster and Power Processor for New Millenium's Deep Space 1 Mission, AIAA Paper 97-2778, July 1997.

30. Andrenucci, M., Marcuccio, S., Genovese, A., Bartoli, C., Gonzalez, J., and Saccoccia, G., FEEP System Study, International Electric Propulsion Conf., Paper 93-156, Sept. 1993.

31. Curran, F. M., Peterson, T., and Pencil, E., Pulse Plasma Thruster Technology Directions, AIAA Paper 97-2926, July 1997.

32. Paccani, G., and Chiarotti, U., Behavior of Quasi-Steady Ablative MPD Thrusters with Different Propellants, International Electric Propulsion Conf., Paper 95-118, Sept. 1995.

33. Toki, K., Shimizu, Y., and Kuriki, K., Experiment (EPEX) of a Repetitively Pulsed MPD Thruster Onboard Space Flyer Unit (SFU), Proceedings of the 25th International EP Conference, Electric Rocket Propulsion Society, Columbus, OH, 1997 (IEPC Paper 97-120).

34. Anon (2017). Delta-v. [image] Available at: https://en.wikipedia.org/wiki/Delta-v#cite_note-marsdeltavs-4 [Accessed 10 Aug. 2017].

35. Anon (2001). Rockets and Space Transportation. [online] Available at: https://web.archive.org/web/20070701211813/http://www.pma.caltech.edu/~chirata/deltav.html [Accessed 10 Aug. 2017].

36. Anon (n.d.). Delta-V Calculator. [online] Strout.net. Available at: http://www.strout.net/info/science/delta-v/intro.html [Accessed 10 Aug. 2017]

37. NASA (2006). Gridded ion thrusters working description. [image] Available at: https://en.wikipedia.org/wiki/Ion_thruster#/media/File:Ion_engine.svg [Accessed 15 Aug. 2017].

38. Goebel, D. and Chu, E. (2011). High Current Lanthanum Hexaboride Hollow Cathodes for High Power Hall Thrusters.

39. Holste, K et al. In Search of Alternative Propellants for Ion Thrusters. 2015. Presentation. Hyogo-Kobe Japan. 2015.

40. Meija, J., Coplen, T., Berglund, M., Brand, W., De Bièvre, P., Gröning, M., Holden, N., Irrgeher, J., Loss, R., Walczyk, T. and Prohaska, T. (2016). Atomic weights of the elements 2013 (IUPAC Technical Report). Pure and Applied Chemistry, 88(3).

41. Goulart, M., Kuhn, M., Kranabetter, L., Kaiser, A., Postler, J., Rastogi, M., Aleem, A., Rasul, B., Bohme, D. and Scheier, P. (2017). Magic Numbers for Packing Adamantane in Helium Droplets: Cluster Cations, Dications, and Trications. The Journal of Physical Chemistry C, 121(20), pp.10767-10772.

42. Polzin, K. Propulsion System Development for The Iodine Satellite (Isat) Demonstration Mission. Hyogo-Kobe Japan. 2015. [Accessed 28 May 2017]

43. Lenzke, K., Landt, L., Hoener, M., Thomas, H., Dahl, J.E., Liu, S. G., Carlson, R. M. K., Möller, T, and Bostedt, C., Experimental determination of the ionization potentials of the first five members of the nanodiamond series. The Journal of Chemical Physics., 127:084320, 2007.

44. Rander, T., Staiger, M., Richter, R., Zimmermann, T., Landt, L., Wolter, D., Dahl, J.E., Carlson, R. M. K., Tkachenko, B. A., Fokina, N.A., Schreiner, P.R., and Möller. T., Bostedt. C., Electronic structure tuning of diamondoids through functionalization. The Journal of Chemical Physics, 138(2), 2013

45. Adhikari, B., and Fyta, M. Towards double-functionalized small diamondoids: selective electronic band-gap tuning. Nanotechnology, 26:035701, 2014.

46. Gunawan, M. A., Hierso, J.C., Poinsot, D., Fokin, A. A., Fokina, N. A., Tkachenko, B. A., and Schreiner, P.R., Diamondoids: functionalization and subsequent applications of perfectly defined molecular cage hydrocarbons. New Journal of Chemistry., 38:28–41, 2014.

47. Kieckhafer, A., King, L.B., Energetics of Propellant Options for High-Power Hall Thrusters, (2005) 41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit.

48. Wirz, R., Sullivan, R., Przybylowski, J., and Silva, M. (2008). Hollow Cathode and Low-Thrust Extraction Grid Analysis for a Miniature Ion Thruster. International Journal of Plasma Science and Engineering, 2008, pp.1-11.